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SUMMARY

In order to provide means for accurate in-flight simulation of V/STOL aircraft, a computer model technique has been adapted to a variable-stability helicopter. Unlike the stability-derivative simulation technique, which is usually employed in variable-stability aircraft, the model approach produces a response which is independent of the dynamics of the test vehicle. The aircraft response, therefore, is a function only of the evaluation pilot's control inputs and the dynamics which are programed into the analog computing equipment.

In-flight time histories of the helicopter response and the corresponding commanded response are presented to illustrate the effectiveness of the technique. The results indicate that the model technique does, in fact, provide a feasible, accurate, and flexible approach to in-flight simulation.

INTRODUCTION

Utilization of the full potential of V/STOL aircraft for IFR operation is not currently possible. The restriction is due in part to a lack of applicable handling-qualities criteria. A useful tool for establishing criteria is an airborne simulator which is capable of providing both a realistic environment and an accurate reproduction of various aircraft dynamics.

As a means of achieving a realistic environment for handling-qualities investigations, the National Aeronautics and Space Administration has modified a modern twin turbine helicopter, which was supplied by the U.S. Army, to provide variable-stability characteristics. The accurate reproduction of aircraft dynamics is achieved through use of the so-called model simulation technique, which has been indicated to be feasible for airborne simulator application by a theoretical study presented in reference 1. Unlike the stability-derivative simulation technique, which is usually employed in variable-stability aircraft, the model technique produces a response which is independent of the dynamics of the test vehicle itself. The aircraft motion, therefore, is a function only of the pilot's control inputs and the dynamics which are programed into the analog computing equipment.

The purpose of this paper is to present a description of the computer model technique of simulation used by NASA in a variable-stability helicopter. Limitations of this technique encountered under operating conditions are discussed. Aircraft response time histories are presented to illustrate the effectiveness of the model technique. A general description of the entire variable-stability system is also included.

DESCRIPTION OF EQUIPMENT

Helicopter

The variable-stability helicopter, shown in figure 1, is powered by twin turbine engines. The "dead man's region" of the height-velocity diagram normally associated with single-engine helicopters is not a limiting factor with this vehicle, thus permitting operation at all altitude and airspeed combinations. The present operating gross weight is 13,000 pounds compared with the maximum gross weight of 15,500 pounds. Therefore, the further addition of other research equipment poses no weight problem. The aircraft's physical characteristics are given in table I.

Figure 1.- Variable-stability helicopter. L-63-8406

TABLE I.- PHYSICAL CHARACTERISTICS OF TEST HELICOPTER

Design gross weight, lb	15,500
Operating gross weight, lb	13,000
Moments of inertia:	
Pitch, slug-ft^2	75,000
Roll, slug-ft^2	9,200
Yaw, slug-ft^2	71,000
Number of blades per rotor	3
Rotor rotational speed, rpm	268
Mechanical control travel:	
Longitudinal, in.	±5.5
Lateral, in.	±3.6
Pedal, in.	±2.3
Collective pitch, in.	12.8
Electric control travel:	
Longitudinal, in.	±6.1
Lateral, in.	±6.7
Pedal, in.	±2.6
Collective, in.	13.8
Control power (hovering):	
Pitch, ft-lb/in.	23,000
Roll, ft-lb/in.	3,500
Yaw, ft-lb/in.	7,800
Unaugmented damping:	
Pitch, ft-lb/rad/sec	38,000
Roll, ft-lb/rad/sec	7,000
Yaw, ft-lb/rad/sec	≈0

Aircraft control moments are produced in the following manner. Fore and aft motion of the center stick produces pitching moments by differential collective pitch on the front and rear rotor. Rolling moments are achieved by lateral motion of the center stick which produces lateral cyclic pitch on both rotors. Differential lateral cyclic pitch on the two rotors in response to pedal inputs produces yawing moments.

The cockpit arrangement shown in figure 2 is standard except for relocation of some instrumentation which was necessitated by the installation of special electronic and navigation equipment. The controls - pitch, roll, yaw, and collective pitch - on the right-hand side of the cockpit were modified to a "fly-by-wire" system. Electrical outputs from these controls command the model aircraft response which is programed into analog computing equipment. When it is desired to operate without computers, the electrical outputs from the controls and from motion sensors may be fed directly into the electrical input servo system to drive the control surfaces. The left-hand controls are unmodified and are continuously

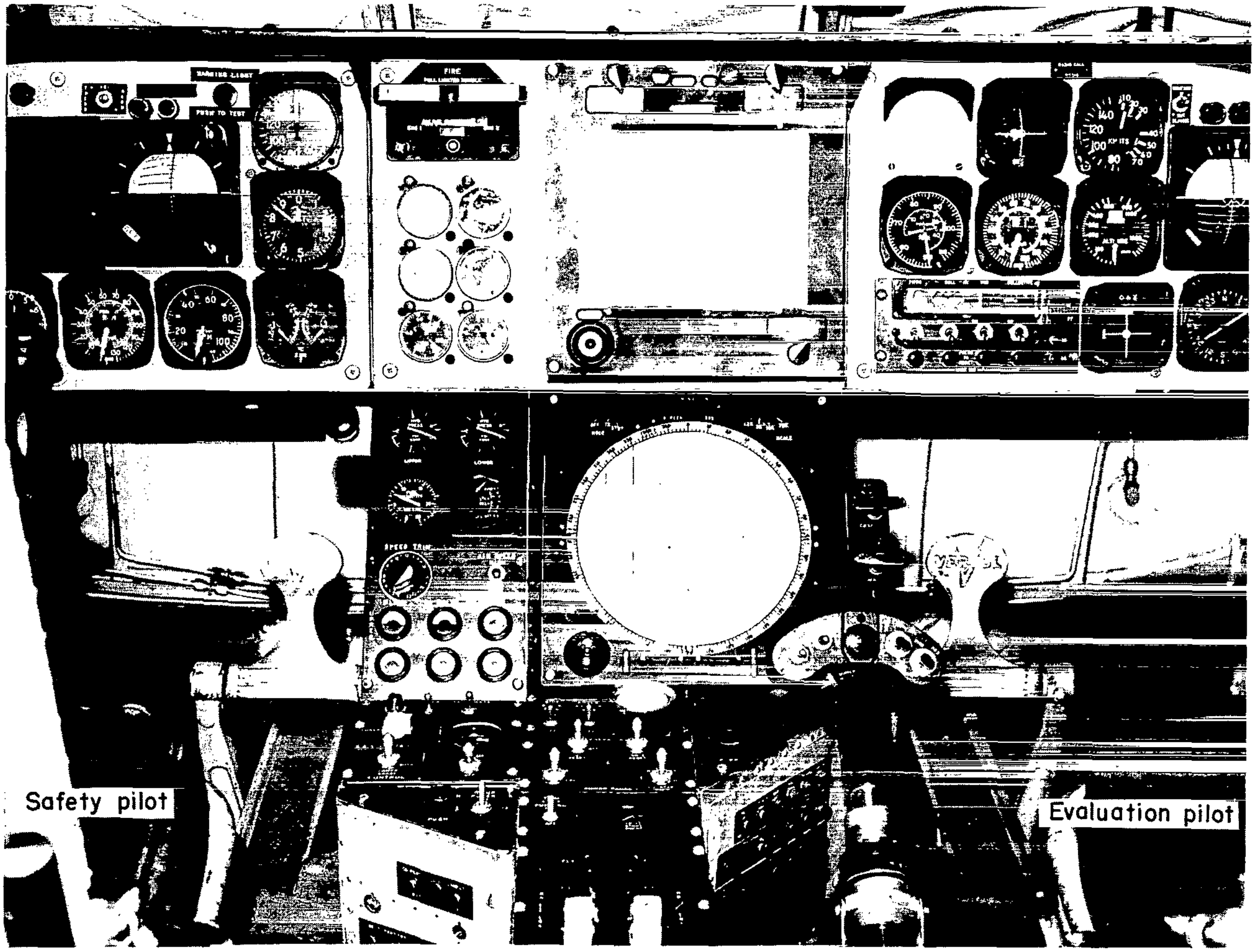

Figure 2.- Cockpit of variable-stability helicopter. L-63-8282

monitored by the safety pilot. Before engagement and after disengagement of the variable-stability system, the safety pilot flys the aircraft through the normal system.

Variable-Stability System

The variable-stability system includes four identical independent axes - pitch, roll, yaw, and collective pitch. A simplified signal-flow diagram is shown in figure 3. Control inputs by the evaluation pilot produce electrical voltages at the signal plugboard. The signal is operated on and shaped by the equations in the analog computer. The resulting signal is routed back through the plugboard to the electrical input servosystem (EISS). The basic function of the EISS is to command a displacement of the control surfaces in direct correspondence to its input voltage. The sensors feed back the subsequent helicopter motions through the plugboard to the computer, where the actual motions are compared with the commanded motions. The safety pilot's control, being mechanically linked to the control surfaces, moves in direct relation to the computer output. In the event of a malfunction or other emergency the safety pilot can take over control by either of two methods. The safety pilot can overpower the EISS input with a 20-pound control force or either pilot can disengage the EISS by pressing a button located on his center stick. The systems components are discussed in detail in the following sections.

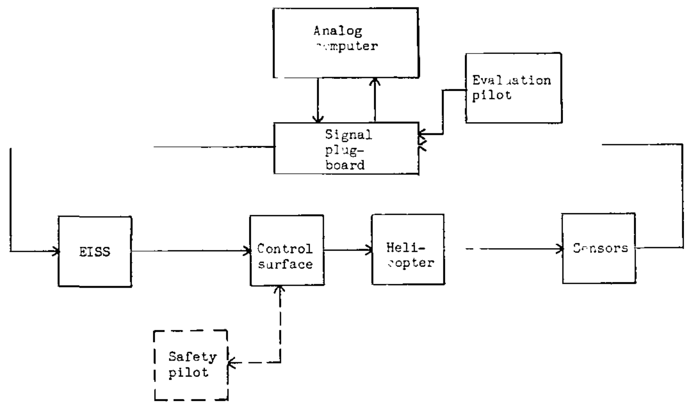

Figure 3.- Block diagram of variable-stability system.

Computing equipment.- The computing equipment, which is shown to the left of the signal plugboard in figure 4, comprises two PACE TR-10 analog computers. The computers are slaved so that both are operated from either computer-control panel. Sufficient computing elements are available for programing the pitch, roll, yaw, and vertical degrees of freedom simultaneously. Nonlinear components such as variable-diode-function generators and comparators are also available. The computers were shock mounted to avoid vibration.

Signal plugboard.- Details of the signal plugboard are shown in figure 5. Outputs from all sensors, in addition to pilot's outputs, are available on the right side of the plugboard. The left side contains input jacks to the EISS. Five input jacks are provided for each axis with a gain potentiometer for each input. Three of these potentiometers are located on the plugboard, and two are located on an overhead panel in the cockpit for added flexibility.

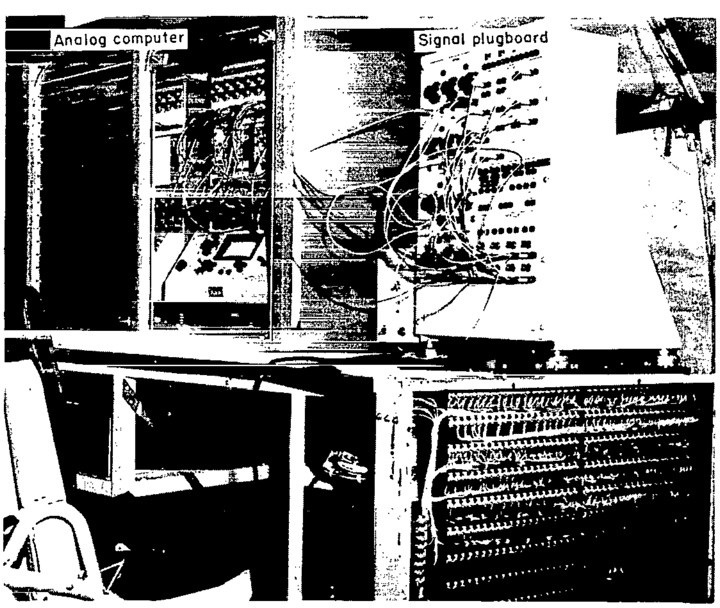

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Figure 4.- Analog computing equipment and signal plugboard.

Electrical input servosystem.- The EISS is a completely transistorized system and weighs less than 75 pounds including actuators and wiring. As previously described, the basic function of the EISS is the conversion of d-c voltage inputs (inputs to the left side of the plugboard) to control-surface displacements. Additional features of the EISS are demonstrated by way of the block

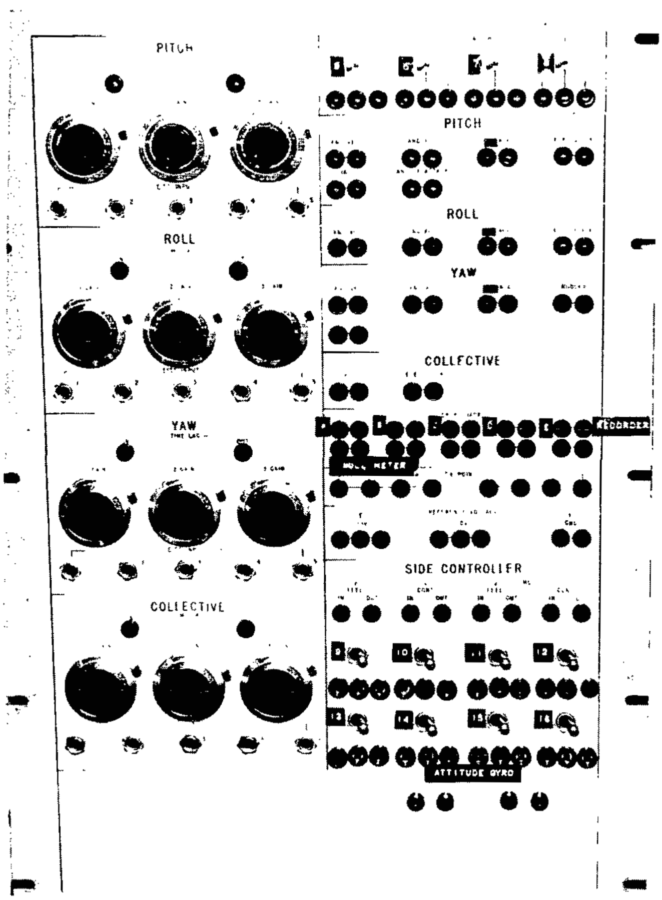

Figure 5.- Signal plugboard. L-63-8283

diagram shown in figure 6. The sum of the EISS inputs for a given axis is fed into the limiting circuit. This circuit limits the maximum control-surface displacement which the signal can command and is used primarily when exploring new configurations and at the beginning of each flight until satisfactory operation of the system is ascertained. A canceling network is provided in the EISS circuit to prevent the possibility of transient or steady-state inputs to the actuator at the instant of engagement of the system. Prior to engagement, the relay remains closed and the canceling network continuously sums the input to the actuator to zero, thereby canceling any inputs to the EISS actuator. The input to the actuator is displayed to the pilot as a safety measure. Upon engagement, the relay automatically opens and the canceling network retains the value which it had at the instant of engagement.

In order to determine the dynamic characteristics of the overall EISS—helicopter-control boost system, frequency-response tests were run. Sinusoidal signals were fed into the EISS and the output was measured at the control surface. The results of these tests can be approximated by a quadratic system with a natural frequency of 15 cps and a damping factor of 0.6. All the axes were essentially identical.

Sensors.- The main sensor package is shown in figure 7. It includes instruments to sense angular velocities, angular accelerations, and linear accelerations relative to the principal inertial axes. Finding a region of minimum vibration was the primary consideration in the installation of this package. When the linear accelerometers are used they are corrected for their displacement from the center of gravity. The angle-of-yaw and angle-of-attack vanes and the airspeed sensor shown in figure 8 are mounted on a nose boom. The boom-mounted sensors operate in the free airstream down to approximately 25 knots. Below this speed the vanes are affected by the rotor downwash.

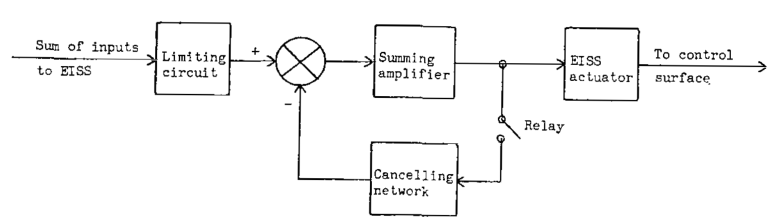

Figure 6.- Block diagram of electrical input servo-system (EISS).

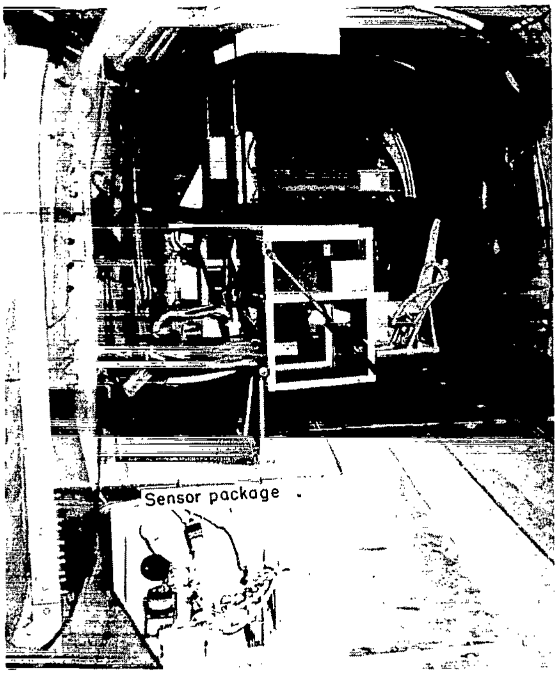

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Figure 7.- Main sensor package installation.

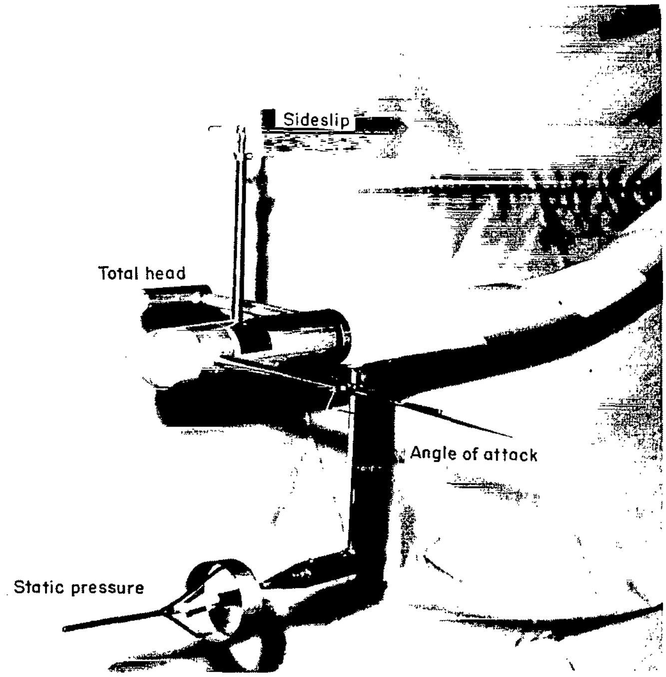

Figure 8.- Boom-mounted sensors. L-63-8281

All sensor outputs, except for the angular accelerometers, are ±10 volts, direct current, corresponding to full scale; the angular accelerometer outputs are ±2 volts.

DESCRIPTION OF SIMULATION TECHNIQUE

General

In an effort to obtain a maximum degree of accuracy and flexibility in the simulation of aircraft dynamics, a computer model approach was adapted to the variable-stability system. In its most elementary form, the model technique is a simple closed-loop servosystem shown in figure 9. In response to the pilot control input, the analog computer generates a signal proportional to the model angular velocity $\dot{\theta}_M$. This signal is fed into the control system and the helicopter responds with an angular velocity $\dot{\theta}_H$. A signal proportional to $-\dot{\theta}_H$ is summed with $\dot{\theta}_M$ to form an error signal E which drives the control system. (Error signals generated from other motion, e.g., attitude, may be used; however, a rate error signal

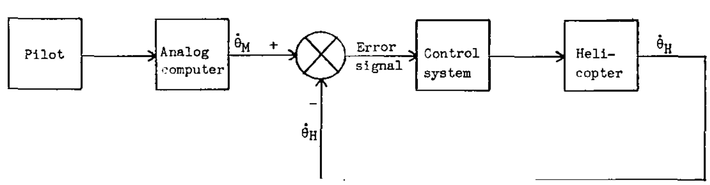

Figure 9.- Simplified model simulation technique.

is preferred in this application.) Whenever $E \neq 0$, the control system drives in the direction necessary to force $\dot{\theta}_H$ to equal $\dot{\theta}_M$. The key to an accurate simulation (i.e., maintaining E near zero) is for the gain on the error-signal loop to approach infinity. The error-signal gain is defined as the angular acceleration which the helicopter generates to cancel a unit error in angular velocity $\left(E \triangleq \frac{\ddot{\theta}_H}{\dot{\theta}_H - \dot{\theta}_M}\right)$. It is noted, therefore, that E has as its unit 1/second and represents the reciprocal of the system time constant. For example, if E = 10/second the helicopter response will lag the model response with a 1/10-second first-order time delay.

It should be noted that $\dot{\theta}_M$ is a function of the pilot control input and any other moments programed into the model. For example, directional stability for operation above 25 knots (below which speed the yaw vane is operating in the rotor downwash) is achieved by feeding a signal from the yaw vane into the yaw axis of the model. In hovering flight, where some of the sensors become unreliable, simulated sensor outputs can be generated by the computer to provide inputs to the model. Additional stability parameters which can be handled in this manner include angle-of-attack stability, speed stability, dihedral effect, and the like. Variations in these procedures may be utilized for conditions between hovering and 25 knots, for example, by the use of ground-speed information.

From the standpoint of producing a truly realistic simulation, it is desirable to have turbulence disturb the model. In forward flight where air direction and velocity-sensor inputs to the model are utilized, the correct disturbance is automatically achieved. In hovering, where simulated sensor inputs are employed, turbulence is not simulated directly; however, for the hovering and low-speed cases turbulence can be generated artificially and fed into the model.

Development of Technique

The preceding discussion indicates the desirability of high gain on the error signal. The error-signal gain is also indicative of the ability of the system to resist or damp unwanted motion produced by external disturbances. Unfortunately, as in other servosystems, system instability limits the maximum gain which can be achieved. In this particular application, amplification of vibrations which are picked up by the sensors imposes an additional gain limitation.

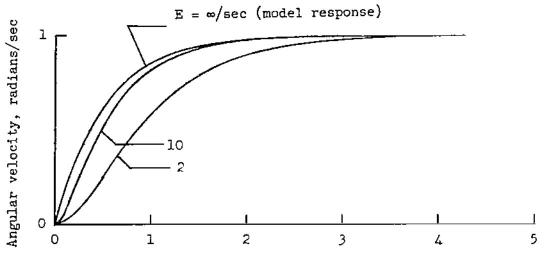

Figure 10.- Effect of error-signal gain on aircraft initial response to control step input.

From the piloting standpoint, the most adverse effect of a limited error gain is the lag (first-order time delay) in the initial response following a control displacement. Computer-generated time histories presented in figure 10 illustrate the lag in the initial response corresponding to error-signal gains equal to 2/second and 10/second. The model response ($E = \infty$/sec) shown is that of a first-order system to a simulated step input.

The lag in the initial response following a control input can be eliminated by the inclusion of a lead network. The broken lines in figure 11 show schematically the relation of the lead network to the system as a whole. By using a lead network, the control system is forced to move simultaneously with the pilot control input. Any remaining lag which still exists in the initial response is a function of the helicopter control system. At any rate, this remaining lag is fairly typical of what might be expected in any vehicle and probably contributes to a more realistic simulation.

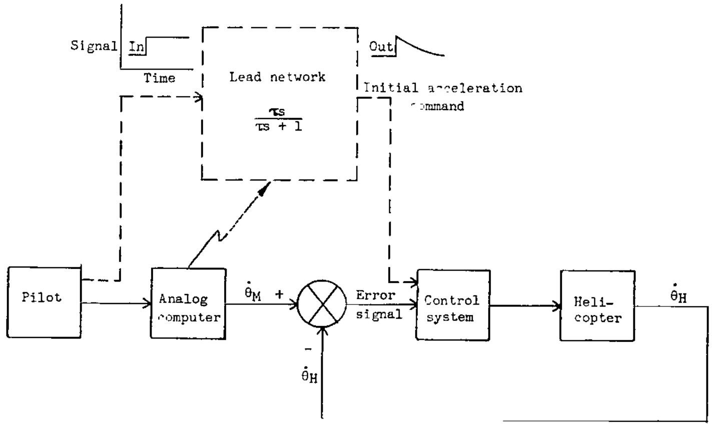

Figure 11.- Addition of lead network to model simulation technique.

After the signal from the lead network has provided the desired initial response, it has no useful purpose. It has been found that a more accurate simulation can be obtained by removing this signal as a function of time. For example, the shaping which the lead network performs on a step input is as shown in figure 11 and has a transfer function $\frac{\tau s}{\tau s + 1}$ where τ is the time constant of the lead network and s is the Laplacian variable. The effectiveness of the lead network in producing an accurate initial response is illustrated by computer-generated time histories in figure 12.

Application of Technique

The error-signal gain was adjusted during hovering flight for each of the axes. With the system engaged, the error-signal gain was slowly increased to the maximum practical value. In the case of the pitch axis, the maximum gain was limited by control-system instability. At a gain of approximately 14.4/second, the pitch-control system began surging with a self-sustained oscillation of approximately 1.5 cps. The gain was subsequently reduced to a satisfactory value of 10.3/second. Changes were then made in the model dynamics and, as was expected, were found to have no effect on the control-system stability.

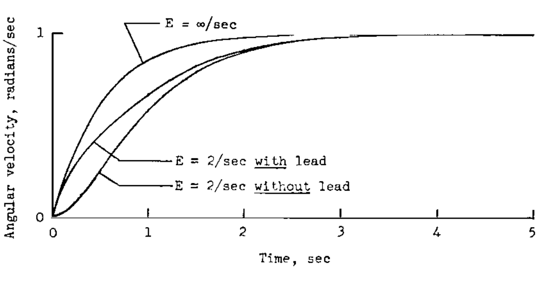

Figure 12.- Effect of lead network in eliminating lag in initial response.

Increases in the gain for the roll and yaw axes were limited, not by control-system instability, but by vibration fed into control system by the angular-velocity sensors. It is recalled that the angular-velocity sensors are in the error-signal loop. Therefore, increasing the error-signal gain serves to amplify the noise which is generated by the helicopter vibration. Isolation of the roll- and yaw-rate sensors from linear vibrations is under consideration. The error-signal gain for the roll and yaw axes was limited to 3.5/second and 2.4/second, respectively.

The gain of the lead network is adjusted whenever the control sensitivity of the model is changed in order to produce identical initial responses of the model and the helicopter. The time constant τ of the lead network was based primarily on an optimization of the control-surface movement. A computer analysis was made in which improvements in the response were weighed against the control-surface motions which were commanded. It was found that large values of τ resulted in a great deal of control overshoot and subsequent hunting action following a control input. An optimum value of τ for the lead network for all axes appeared to be about 0.2 second, for which value the "lead" curve in figure 12 was generated.

Theoretical calculations indicate that tuning of the lead network is important if a low maximum error-signal gain and high inherent stability of the aircraft occur in the same axis. Basically the tuning involves canceling only a portion of the pilot's input to the lead network. This augments the noninfinite error-signal gain in washing out the basic helicopter stability. Tuning was not an important consideration in the present application.

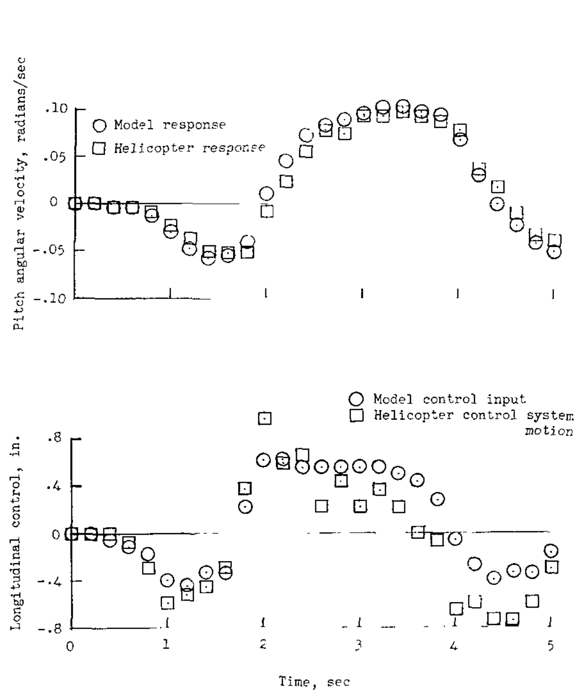

Figure 13.- Comparison of helicopter pitching angular velocity response and commanded response for control input to model.

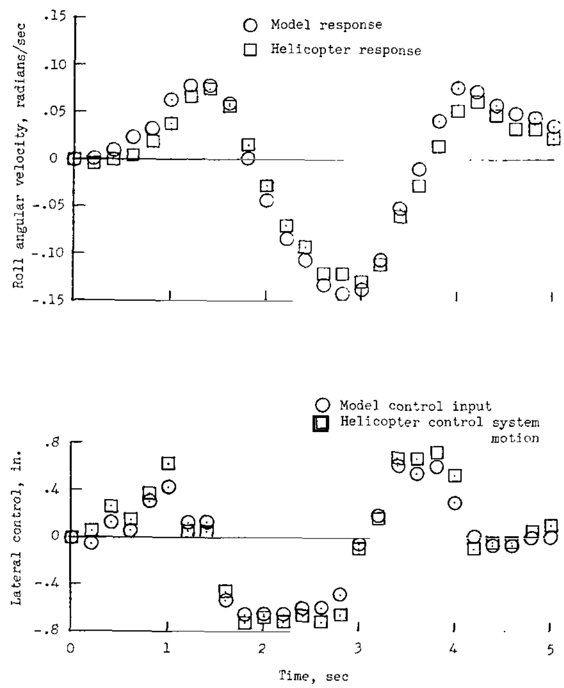

Figure 14.- Comparison of helicopter rolling angular velocity response and commanded response for control input to model.

RESULTS

In-flight time histories shown in figures 13 to 15 illustrate the effectiveness of this technique in pitch, roll, and yaw. These time histories were obtained while hovering; the circles on the top curve of each figure represent the angular velocity of the model in response to a control step input which is represented by the circles on the bottom curve of the figure. The square symbols on the top curve of each figure represent the helicopter angular veloctiy as commanded by the model. The square symbols on the bottom curve of each figure represent the motion of the safety pilot's control which is driven by the computer output and is indicative of the control-surface motion. It may be seen from figure 15 that even in the yaw axis, which has the lowest error-signal gain, the simulation is fairly accurate.

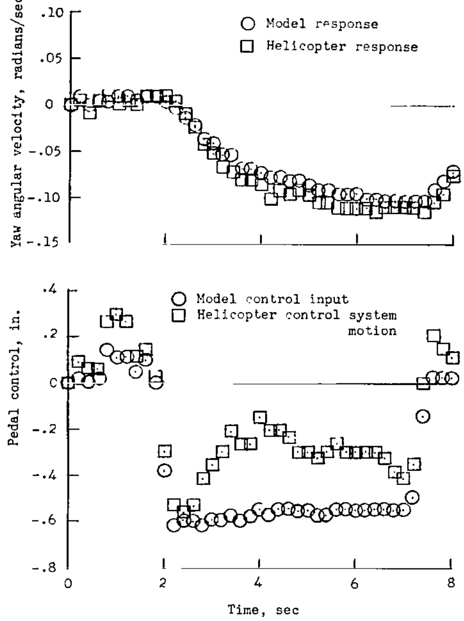

Figure 15.- Comparison of helicopter yawing angular velocity response and commanded response for control input to model.

In order to determine more fully the effectiveness of the simulation technique, standard static directional stability tests were made in level flight at 60 knots. This test is normally used to determine the static directional stability of an aircraft, that is, its tendency to yaw into the relative wind, and is performed in the following manner. The pilot applies the necessary pedal control to sideslip the aircraft slowly, first in one direction and then the other. The slope of the pedal-position—sideslip-angle curve provides a measure of the static directional stability. In figure 16, the results of such a test are shown with positive static directional stability in the computer model. Calculations based on the slope of this curve indicate that the desired stability was achieved within 3 percent. The scatter is caused in part by the effect of turbulence both on the vane which senses sideslip, and on the aircraft itself. The static directional stability of the basic helicopter, which is both variable and slightly unstable, is shown for comparison in figure 17.

During the course of research flight utilizing this technique, large changes were successfully made in various stability and control derivatives, including angular acceleration per inch of control motion, angular-velocity damping, and several static-stability derivatives. No practical limitation in the simulation equipment has yet been encountered for these quantities. It appears, therefore, that the stability characteristics of a wide variety of aircraft types can be simulated. A limitation does exist with respect to the maximum angular-acceleration capability of the aircraft, which is given in the following table:

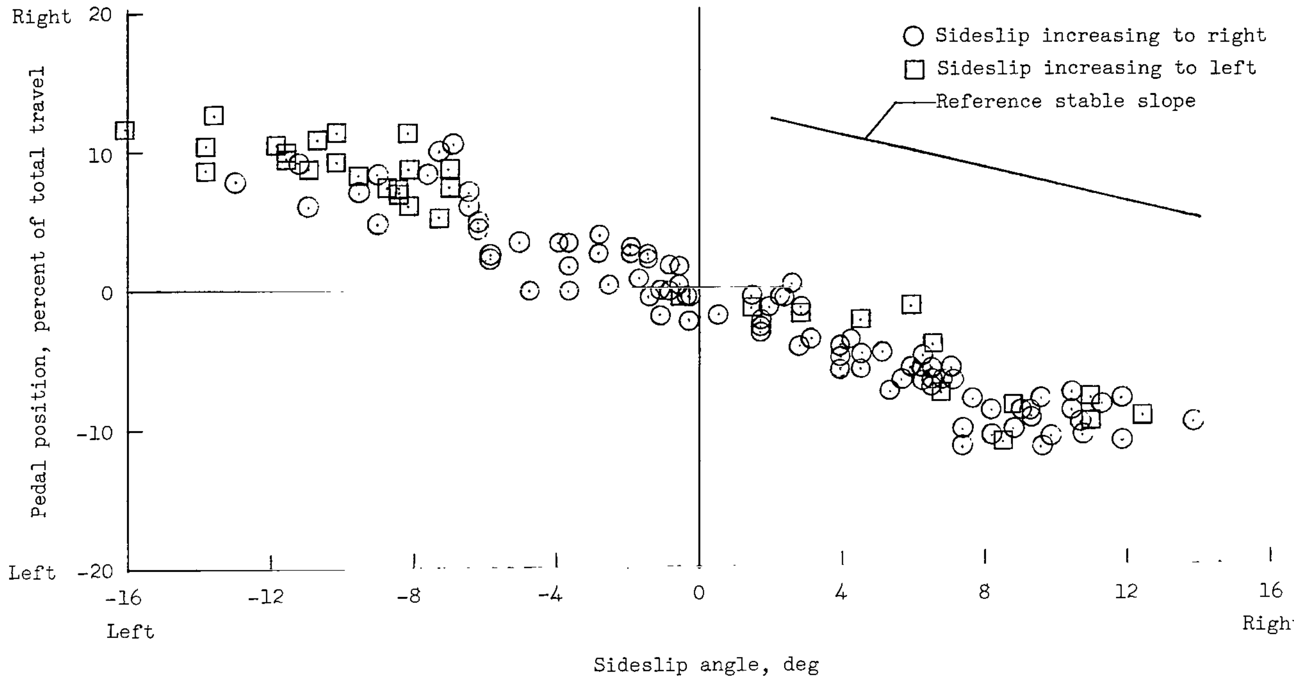

Figure 16.- Apparent aircraft directional stability produced by model.

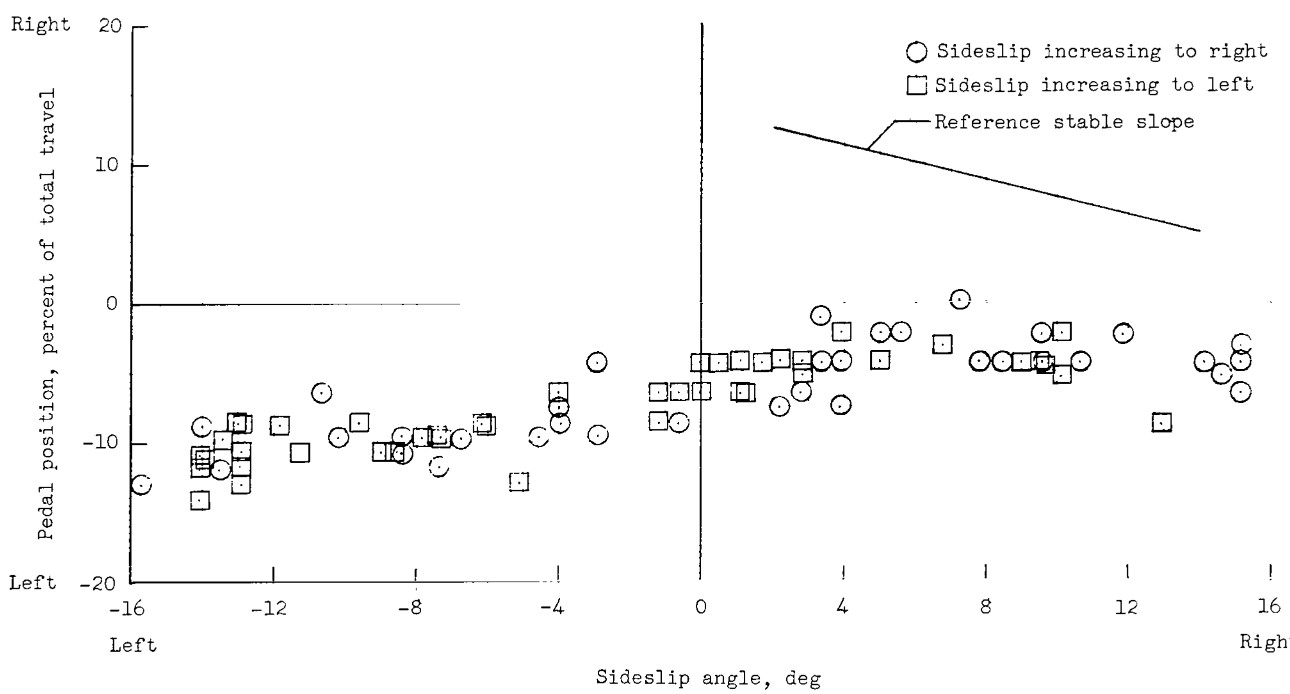

Figure 17.- Directional stability of basic helicopter.

Axis, radians/sec^2	Maximum angular acceleration
Pitch	1.7
Roll	1.3
Yaw	0.25

CONCLUDING REMARKS

A model-controlled simulation technique has been adapted to a relatively sophisticated variable-stability helicopter for study of low-speed handling-qualities requirements. The ability of the technique to wash out the stability of the basic helicopter and thus to command the computed response has been demonstrated by step inputs while hovering and by static stability tests in forward flight. Some lag problems were encountered because of certain limitations on the maximum error-signal gain which could be achieved. However, these problems were largely overcome by introduction of a lead network which produces the correct initial response following control inputs. The results indicate that the model technique does, in fact, provide a feasible, accurate, and flexible approach to in-flight simulation.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., October 17, 1963.

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1. Gould, D. G.: The Model Controlled Method for the Development of Variable Stability Aircraft. Aero. Rep. LR-345 (N. R. C. No. 6914), Nat. Res. Council of Canada (Ottawa), June 1962.